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Design Challenges of Power Systems for Instrumented Spacecraft with Very Low Perigees in the Earth's Ionosphere

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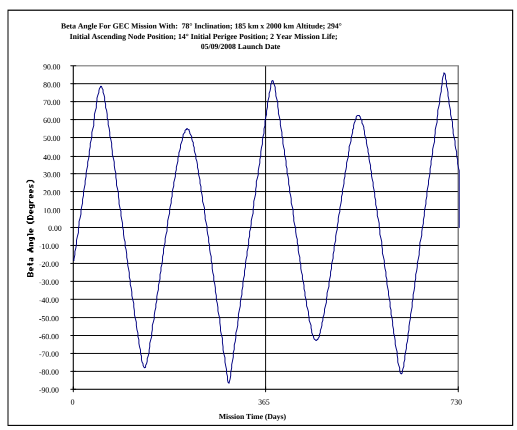
ABSTRACT

Designing a solar array to power a spacecraft bus supporting a set of instruments making in situ plasma and neutral atmosphere measurements in the ionosphere at altitudes of 120km or lower poses several challenges. The driving scientific requirements are the field-of-view constraints of the instruments resulting in a three-axis stabilized spacecraft, the need for an electromagnetically unperturbed environment accomplished by designing an electrostatically conducting solar array surface to avoid large potentials, making the spacecraft body as small and as symmetric as possible, and body-mounting the solar array. Furthermore, the life and thermal constraints, in the midst of the effects of the dense atmosphere at low altitude, drive the cross-sectional area of the spacecraft to be small particularly normal to the ram direction. Widely varying sun angles and eclipse durations add further complications, as does the growing desire for multiple spacecraft to resolve spatial and temporal variations packaged into a single launch vehicle. Novel approaches to insure adequate orbit-averaged power levels of ~250W include an oval-shaped cross section to increase the solar array collecting area during noon-midnight orbits and the use of a flywheel energy storage system. The flywheel could also be used to help maintain the spacecraft's attitude, particularly during excursions to the lowest perigee altitudes. This paper discusses the approaches used in conceptual power designs for both the proposed Dipper and the Global Electrodynamics Connections (GEC) Mission currently being studied at the NASA/Goddard Space Flight Center.

ORBIT MECHANICS

The higher the inclination of the orbit is, the greater the global coverage for the situ instruments. Unfortunately, the high inclination results in a widely varying beta angle. The beta angle is defined as the angle between the sun line and the orbit plane. A plot of the beta angle is shown below over 2-years for a mission with a

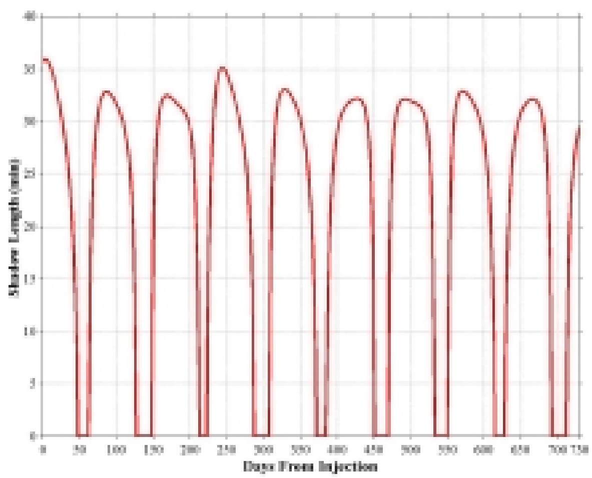


185 x 2000 km orbit inclined to 78°. The beta angle should not change significantly when the spacecraft is dipping to 120-130 km. The simulation initial conditions were: May 9, 2008 launch and 1:36 a.m. initial ascending node. The GEC mission was planned assuming a 185-200km x 2000km parking orbit and dipping campaigns at areas of interest to 120km x 2000km altitudes.

The solar array has to be designed such that sufficient area is illuminated to provide the required power with the sun moving a possible ±90° with respect to the orbit plane.

For missions where the solar array must be body-mounted and a solar array drive cannot be used to track the moving sun, this causes the physical area of the solar array to be much larger and the cost of the cells to be much higher.

The beta angle and the position of perigee, drive the eclipse duration, another key parameter for designing the solar array. The initial conditions described above and an initial perigee local time of 05:00 and argument of perigee of 80° (perigee latitude is at 75° N) produce the following eclipse duration profile. The full-sun periods are 2-3 weeks in length. The eclipse periods are approximately 9 weeks in length. Unfortunately, the initial conditions chosen start the mission at the worst shadow condition. Regardless of the initial conditions, the shadow durations will remain the same - only the profile will change over time.



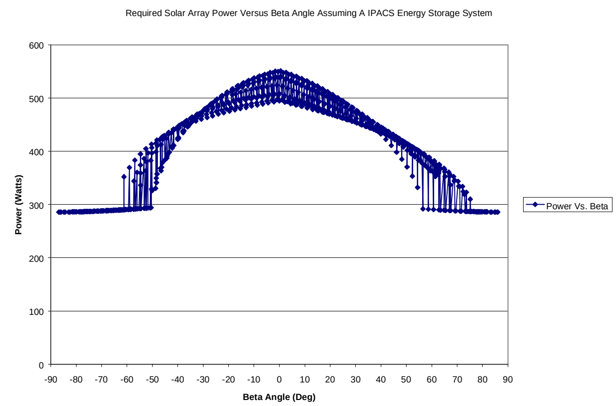
Power Required From The Solar Array Using A Conventional Battery and Momentum Wheel System

The power required from the solar array during the day lit portion of the orbit is about 1.22 times the required load power for the full sun orbits (high beta angle) and about 2.39 times the required load power for the maximum eclipse orbits (low beta angle) assuming a conventional battery for energy storage and an unregulated Direct Energy Transfer (DET) type of power system. The power system was sized assuming 233W.

Power Required From The Solar Array Using A Integrated Power and Attitude Control (IPAC) System Incorporating Inertial Energy Storage Flywheels

A preliminary trade study shows that an IPAC system offers advantages over a conventional battery-momentum wheel implementation.

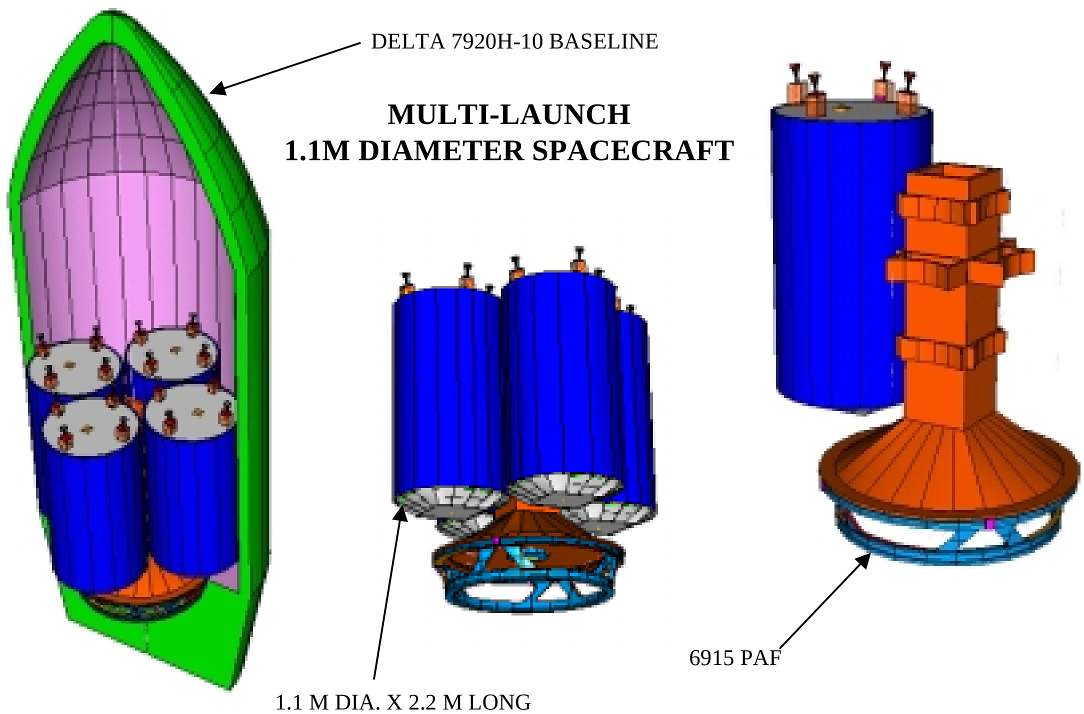
Use of a flywheel type energy storage device reduces the requirement of power from the solar array to 2.10 times the required load power for the maximum eclipse orbits (a 12% reduction). This is because the flywheel is more efficient at "charging". A conventional battery requires taper and trickle charge during which most of the available solar array power cannot be used. The flywheel is also thought to be more efficient in discharge (goal of 90%) than a conventional battery, which is only about 82% efficient in this mode.



The IPACs system is comprised of a high-speed energy storage rotor, an efficient motor/generator, and magnetic bearings. A preliminary design for the rotors required for GEC is two counter rotating wheels each approximately 100mm tall and 350mm in diameter weighing less than 12kg. The pair could provide the energy storage requirement (155Wh) operating between 15,000 RPM and 30,000 RPM. Operating either one of the counter-rotating wheels at 50,000 RPMs would provide the momentum bias (130nms). GEC has funded two feasibility studies that will look into the IPACs option in more detail. The objective is to design, build and test a reaction wheel which meets attitude control requirements while reducing disturbances, in a package suitable to a wide range of missions; examine wheel configurations which maximize reliability and performance, while minimizing mass and volume; and finally, examine advanced approaches to further improve performance with an Integrated Power and Attitude Control System (IPACS). The objective is to build and test a prototype in FY01, build a flight system in FY03, and possibly perform a flight experiment in FY04.

Spacecraft Conceptual Design

The initial conceptual mechanical design for the spacecraft was a cylinder. The objective was to package as many spacecraft in the Delta 7920H-10 as possible, meet the power requirements with a body-mounted solar array, keep the cross-sectional area in the ram direction as small as possible, and keep the length of the spacecraft as close to the diameter as possible. The first cut produced a design of 4 spacecraft with a diameter of 1.1m and a length of 2.2 m. Given these dimensions a study was performed to determine the power available to the instruments assuming a battery for energy storage and then a flywheel for energy storage.

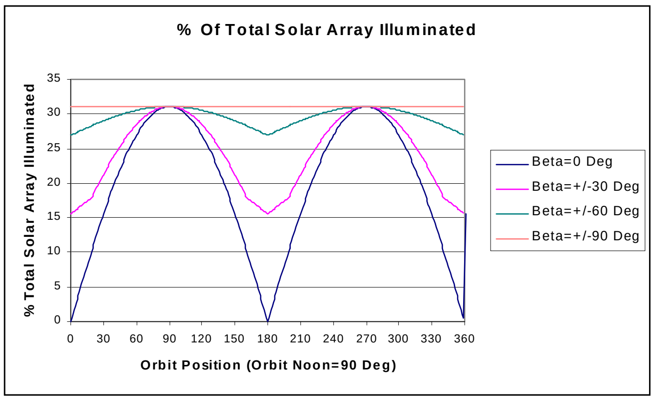


The on orbit configuration of the spacecraft is shown below. The spacecraft is three axis stabilized and nadir pointed. The velocity vector is parallel to the axis of symmetry and the face in the ram direction is slightly beveled to reduce atmospheric drag. The sun angle for any facet of the surface excluding the ends is

$$\cos(\theta_{sun})=\cos(\beta)*\cos(\theta_{orbit}-90°)*\cos(\theta_{facet})-\sin(\beta)*\sin(\theta_{facet})$$

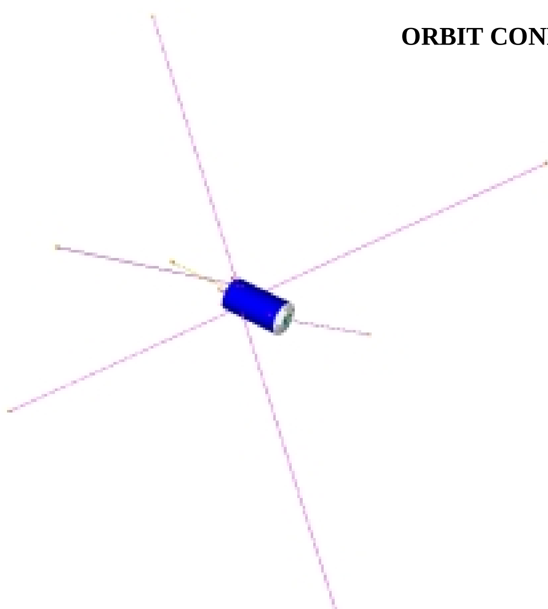
β=angle between the sunline and the orbit plane
θ_{orbit}=angular position of spacecraft in orbit; 90°=orbit noon; 0°=6am; 180°=6pm; 270°=midnight.
θ_{facet}=angular position of panel around spacecraft; 0°=facet normal to sun at orbit noon, β=90°.

Solar Array Predicted Performance

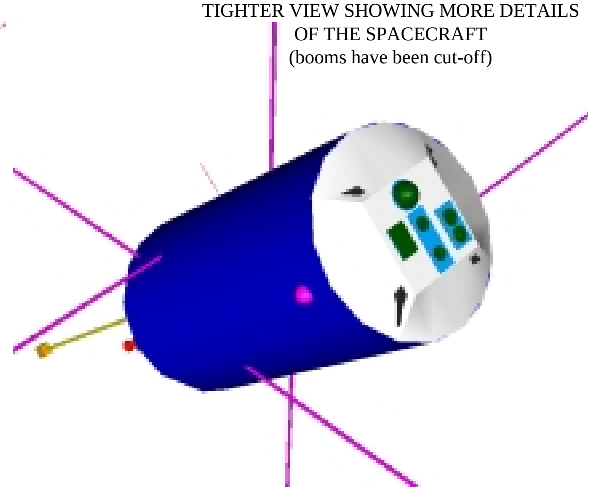


To minimize the size of the solar array the highest efficiency GaAs solar cell was baselined. Triple Junction GaAs with a BOL, bare cell efficiency of 24% was assumed. We assumed that the average efficiency dropped to 22.9% after the cell was covered with a coverglass and assembled onto a panel. This efficiency should also allow margin for measurement uncertainty. For prediction of on orbit performance, an orbit average solar array temperature of 0°C was used for the Beta=0° orbits and an orbit average temperature of 50°C was used for the Beta=90° orbits. The cosine losses were as assumed above with an additional 15% loss for the packing factor and 5% for appendiges using panel real estate. An additional 5% shadowing loss was also assumed. A rough estimate of the loss due to ultraviolet radiation, thermal cycling, and charged particle radiation was done as a function of time. The loss used at the end of 2 years due to ultraviolet radiation was 2.3%, due to thermal cycling was 3.2%, and due to charged particle radiation was 17%. The intensity was varied appropriately for the season.

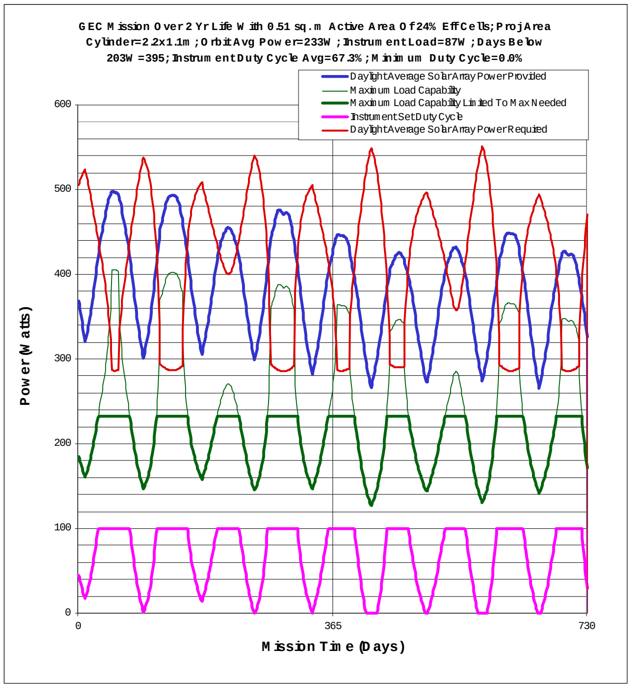
ORBIT CONFIGURATIOM



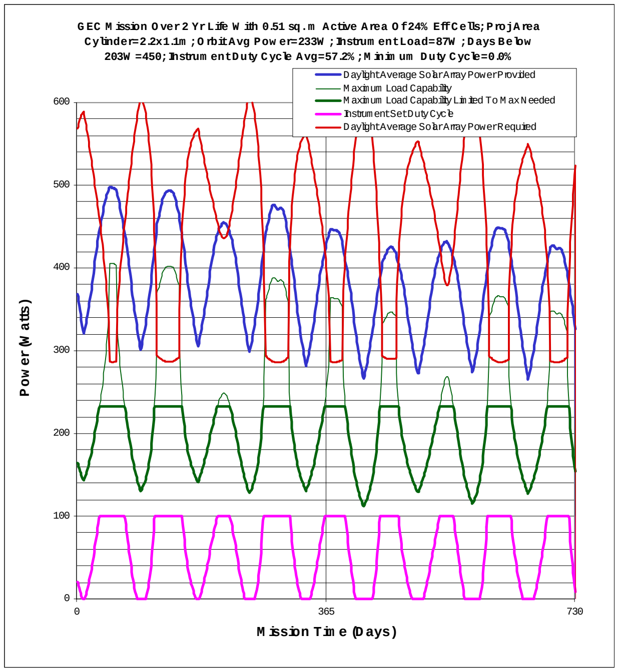
TIGHTER VIEW SHOWING MORE DETAILS OF THE SPACECRAFT
(booms have been cut-off)



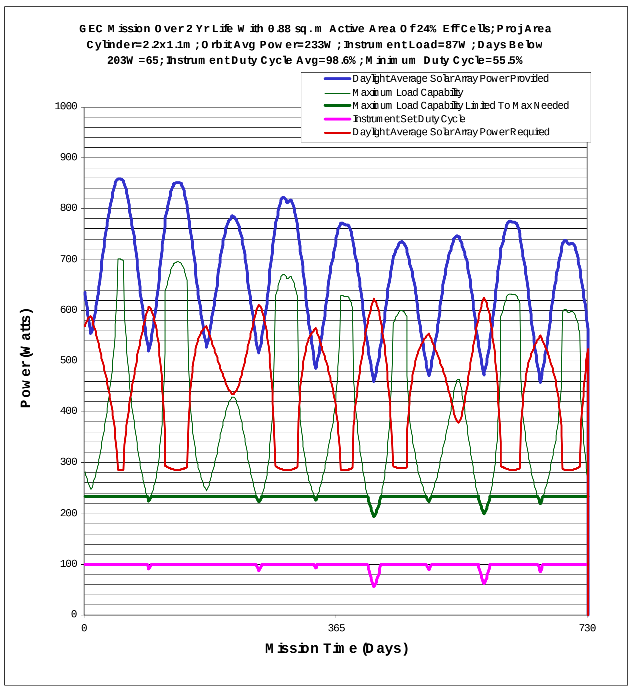
The average power expected from the solar array during the day was calculated for each day of the mission dependent upon the beta angle and eclipse duration for that day and plotted against the requirement to assess the sizing and margins.



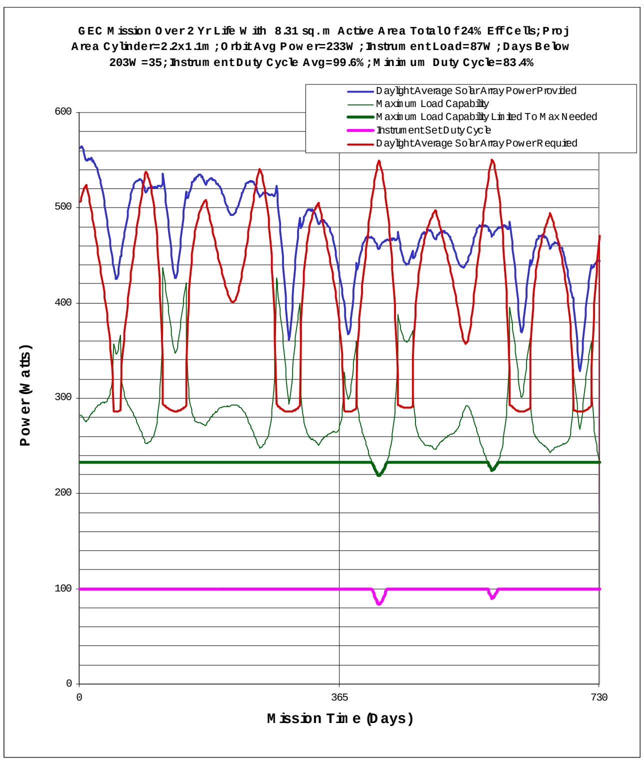
Using the above assumptions, there is not enough area on the 1.1m diameter x 2.2m long spacecraft to support 233W continuously. Duty cycling of the load would have to be performed during the times when there is negative margin. The approximate percentage of duty cycling is about 67.3% average and there are times when no science instruments can be turned on. For a system with a battery, the duty decreases to 57.2% average.



In order to increase the duty cycle of the science instruments to almost 100%, and keep the circular cross-section (~.95m^2), the spacecraft needs to be much longer. For ~98% duty cycle, the length needs to increase from 2.2m to 3.8m (total active area increase from 6.1m^2 to 10.6m^2). This is extremely inefficient because there are periods where as much as 570W of power is wasted.



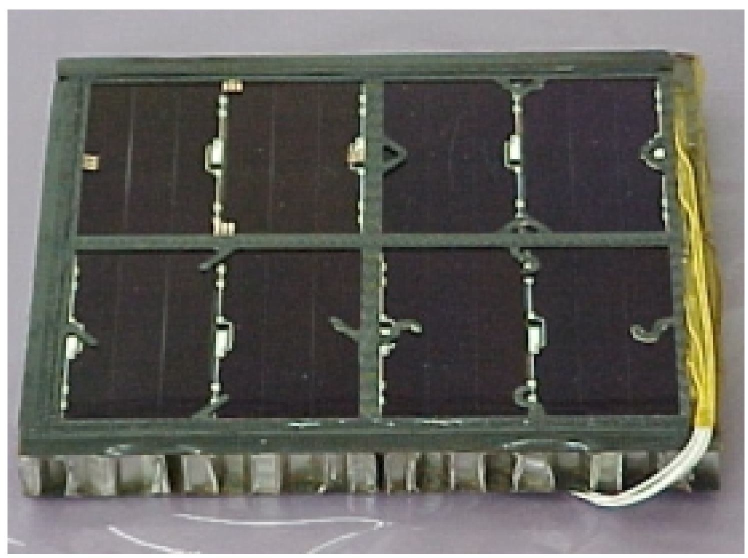
One idea for improving on the body-mounted design is to make the spacecraft oval shaped, increasing the area projected to the sun in the noon-midnight orbits, where the power required is higher, and decreasing the area projected to the sun in the dawn-dusk orbits. To get a preliminary assessment of the advantage of the oval shape, the power margins were calculated for an oval 1.85m across the major axis and .79m across the minor axis. The cross-sectional area is ~ .85m^2 which is less than the cross-sectional area of the 1.1m diameter cylinder (~.95m^2). The total active area is about 8.3m^2 , 21.6% less than the baseline, where the active area was 10.6m^2. The oval offers the additional advantage of allowing the spacecraft to be shorter (2.2m vs. 3.8m) and closer to the diameter. In the next part of the design phase, the feasibility of packaging the oval shape within the fairing whose dimension are 1.85m x .79m in lieu of the 1.1m x 1.1m presently packaged.



In addition to the challenges of providing power with limited area to mount cells and unfavorable sun conditions, the electrostatic cleanliness requirements and environment resulting from dipping into the atmosphere at 120km are also challenging.

Electrostatically Clean Solar Array

An electrostatically clean solar array has less than 0.1 volt across any two points on the array. In addition, there are no exposed potentials or insulators. The traditional methods of accomplishing this have been to use coverglasses with a conductive coating that are interconnected with straps or bond wires and grounded. Between the cells, the interconnects and insulator are covered with layers of an insulative and conductive epoxy. FAST, the last solar array built for GSFC with these cleanliness requirements, was enormously expensive due to the large number and fragility of parts assembled on the array. NASA/GSFC has initiated a new technology demonstration effort with Composite Optics Inc. (COI) to demonstrate the manufacture of an inexpensive and reliable electrostatically clean solar array. In addition to reducing the parts count, the effort is designed to reduce the effect of ESD cleanliness on power output. COI is accomplishing this by using a single aperture over the panel, which is conductive on the exposed side and insulative on the cell side. The aperture covers all parts excluding the cell. The cell has a traditional coverglass which is coated with indium titanium oxide (ITO). This system provides resistance to high temperature, atomic oxygen (AO), and ultraviolet (UV). COI has built a qualification panel, shown below, and is running it through thermal cycling testing. In addition a full scale prototype panel will be built. The final results expected in February 2000.



Qualification Panel with four interconnected configurations (Be-Cu, diamond, slant, serpentine)